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International Space Station Nickel-Hydrogen Battery Start-Up and Initial Performance

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INTERNATIONAL SPACE STATION NICKEL-HYDROGEN BATTERY START-UP AND INITIAL PERFORMANCE

Fred Cohen/Boeing Co. Penni J. Dalton/NASA Glenn Research Center

ABSTRACT

International Space Station (ISS) Electric Power System (EPS) utilizes Nickel-Hydrogen ($Ni\text{-}H_2$) batteries as part of its power system to store electrical energy. The batteries are charged during insolation and discharged during eclipse. The batteries are designed to operate at a 35% depth of discharge (DOD) maximum during normal operation.

Thirty eight individual pressure vessel (IPV) $Ni\text{-}H_2$ battery cells are series-connected and packaged in an Orbital Replacement Unit (ORU). Two ORUs are series-connected utilizing a total of 76 cells, to form one battery. The ISS is the first application for low earth orbit (LEO) cycling of this quantity of series-connected cells.

The P6 Integrated Equipment Assembly (IEA) containing the initial ISS high-power components was successfully launched on November 30, 2000. The IEA contains 12 Battery Subassembly ORUs (6 batteries) that provide station power during eclipse periods. This paper will describe the battery hardware configuration, operation, and role in providing power to the main power system of the ISS. We will also discuss initial battery start-up and performance data.

1.0 INTRODUCTION

At Assembly Complete, the ISS EPS will be powered by 24 batteries during eclipse and extended operation periods. The battery (see Fig. 1) is designed to operate for 6.5 years with a mean-time-between-failure (MTBF) of 5 years when run in the reference design 35% DOD LEO regime. Typical expected discharge currents can range from <25 Amps in a low-demand orbit to as high as ~75 Amps to meet short peaking load requirements at a battery operating voltage range of 76 to s123 Vdc. The ORUs are individually fused to protect the ISS EPS from fault propagation that could result from a cell-to-EPS ground event. Primary charge control is accomplished by a pressure temperature algorithm that incorporates acceptance test data to initialize basic reference parameters.

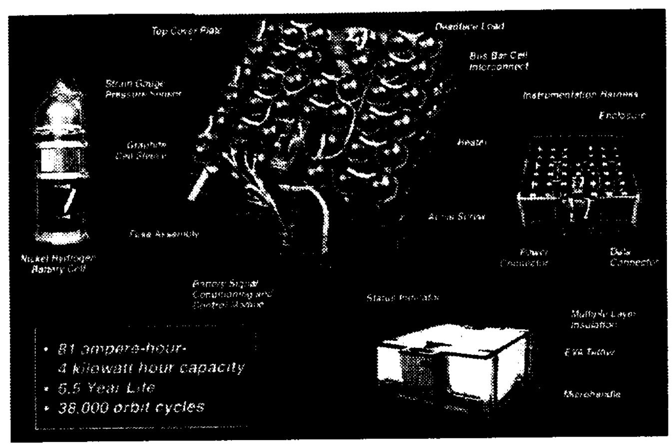

Figure 1. ISS Battery Subassembly ORU

Table 1. Reference Orbit Design Parameters Per Battery Subassembly ORU

Condition	Time (min) Start	Time (min) End	Energy (Watt-hrs)	Power (Watts)
CONTINUOUS POWER REQUIREMENTS				
Constant Power Charge	0.0	43.9		1995*
Taper Charge	43.9	57.0		
Total Charge			1677*	
Constant Power Discharge	57.0	92.0	1342	2300
PEAKING POWER REQUIREMENTS				
Constant Power Charge	0.0	7.5		1554*
Constant Power Charge	7.5	43.9		2072*
Taper Charge	43.9	57.0		
Total Charge			1677*	
Constant Power Discharge	57.0	84.5	967	2110
Constant Power Discharge	84.5	92	375	3000
Total Discharge			1342	
CONTINGENCY POWER REQUIREMENTS				
Constant Power Discharge	0.0	92.0	997*	650
*Designates a maximum value				

The ISS power system is the first on-orbit use of such a large quantity of series-connected IPV Ni-H_2 battery cells (38/76), in an ORU/Battery configuration. Previous ground testing had been performed on 22 IPV NiH_2 cells in series [1]. Therefore, during the ISS program development stage, it was important to demonstrate that the "as-designed" battery could be successfully run (see Table 1). This was accomplished at the Power Systems Facility (PSF) Laboratory at NASA Glenn (then NASA Lewis) Research Center in Cleveland, Ohio in 1992 [2].

2.0 BACKGROUND: INITIAL BATTERY PERFORMANCE TEST SUMMARY

Two Space Station Engineering Model (EM) ORUs were initially tested using an orbital rate capacity (ORC) test, as well as individually LEO cycled at the 35% DOD reference orbit to provide baseline characteristics. After completion of the baseline testing, the hardware was configured as a "battery" by connecting them in series and subsequently running them for 3,000 simulated ISS reference design cycles at a recharge ratio (RR) of 1.043 (as described in Table 2). The ISS design power requirements are specified in units of Watts and, therefore, the cycle regime is power based. The 3,000 "peaking" cycles (see Tables 1 and 2) were performed using the maximum discharge power delivery requirement and a recharge regime that incorporates a taper charge that reduces charging stress at high states of charge (SOCs). The test was performed while maintaining the cell sleeve temperatures at 5 ± 5°C.

Table 2. ISS Simulated Peaking Reference Design Orbit, ≤35% DOD, 1.043 RR

Charge	**57.0 Minutes (total)**
3,108 Watts	7.5 minutes
3,746 Watts	36.4 minutes
3,746 taper to 700 Watts	13.1 minutes
Discharge	**35.0 Minutes (total)**
4,220 Watts	27.5 minutes
6,000 Watts	7.5 minutes

Following completion of 3,000 cycles, the ORUs were subjected to individual orbital rate capacity tests to determine any degradation in performance.

The result is that the ORUs exceeded the ISS design requirements for electrical performance, heat generation, thermal uniformity, and charge management.

3.0 ORU DESIGN CONSIDERATIONS

Remembering that the original ISS battery design effort began in 1988, a long-life, high-performance battery was needed. Therefore, state-of-the-art Ni-H_2 IPV chemistry was chosen at that time, and designed to meet the following ORU requirements:

- 6.5-year design life
- 81-Amp-hr nameplate capacity to limit the maximum reference DOD to less than 35%
- Contingency orbit capability consisting of one additional orbit at reduced power after a 35% DOD without recharge
- 5-year MTBF
- Easy on-orbit replacement using the ISS robotic interface

The cells selected for use in the Battery ORUs are manufactured by Eagle Picher Industries. The cells are RNH-81-5 EPI IPV NiH_2, and utilize a back-to-back plate configuration. They are activated with 31% potassium hydroxide (KOH) electrolytes. The ORUs are assembled and acceptance tested by Space Systems/Loral.

4.0 ISS BATTERY CONFIGURATION

The Battery Subassembly ORU, as designed and built, is pictured below in Figs. 2 and 3.

The NiH_2 cells for the current 12 ISS Battery ORUs were manufactured 3.5 to 4.4 years before the November 30, 2000 launch date. The flight ORUs were used for IEA systems ground testing and final checkout, but were stored open-circuit, discharged, and at −10 °C when they were not in use.

The 12 Battery ORUs were integrated onto the P6 IEA in July 2000 at the Kennedy Space Center (KSC). Two ORUs in series form one battery, for a total of 76 cells in series. These 12 ORUs form six separate batteries, with three batteries on each of two power channels. For the P6, these power channels are designated as 2B and 4B. During insolation, power is supplied to the source bus by solar arrays that meet the demand for user loads, as well as battery recharging. The batteries, through a Battery Charge/Discharge Unit (BCDU), provide the power to the source bus for the ISS during eclipse periods.

Each ORU contains a Battery Signal Conditioning and Control Module (BSCCM). The BSCCM provides conditioned battery monitoring signals from the ORU to the Local Data Interface (LDI) located within the BCDU. Available data includes 38 cell voltages, four pressure (strain gauge) readings, six cell and three baseplate temperatures and are provided as an analog multiplexed voltage. A separate signal provides ORU total voltage output. The BSCCM also accepts and executes commands from the BCDU/LDI to control ORU cell heater and letdown functions.

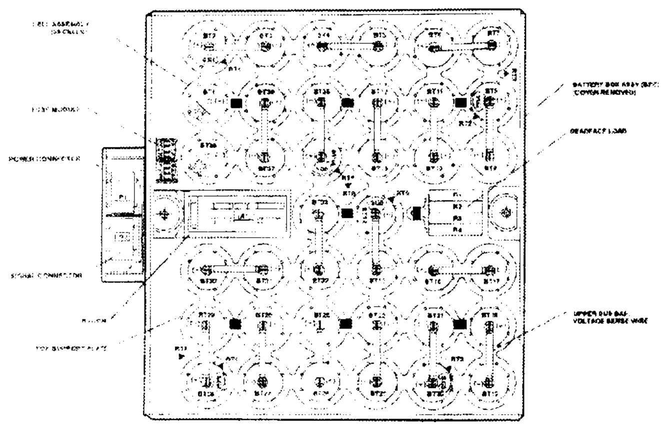

Figure 2. Baseplate Layout - ISS Battery Subassembly ORU

For battery charging, the BCDU conditions power from the source bus and charges the battery at charge setpoints as calculated from the charge algorithm (reference paragraph 6.0). During periods of eclipse, the BCDU extracts power from the battery, conditions this power, and supplies power to the source bus.

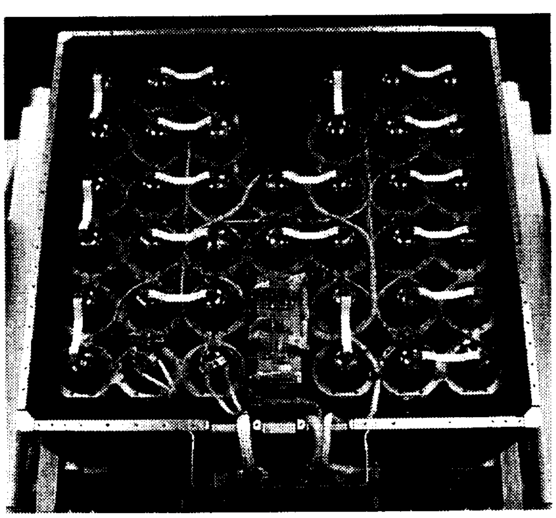

Figure 3. ISS Flight Model Battery Subassembly ORU with Cover Removed

The batteries are actively cooled using the ISS Thermal Control System (TCS). The battery cells are assembled in an ORU box, using a unique finned radiant heat exchanger baseplate. The baseplate is then mounted on the IEA using ACME screws and mated to the TCS. The TCS was designed to maintain the Battery ORUs at a nominal operating temperature range of 5 ± 5°C (41 ± 9°F) with minimum heater operation when run at a 35% DOD LEO regime.

5.0 ISS ON-ORBIT START-UP

The ISS batteries are launched in a discharged state. As a result a multi-orbit start-up was necessary to begin orbital operation. Battery charging was not begun until after solar array deployment and thermal conditioning. System control and operational power was supplied by the National Space Transportation System (NSTS) Auxiliary Power Control Unit (APCU). As a result of the limited capability of this power source and the desire to quickly charge the batteries to 100% SOC, heater operation and battery discharge were inhibited during eclipse.

After thermal conditioning, which consisted of warming the ORUs using their internal heaters to nominal operating temperature (between 0 and 10°C), battery charging was initiated using an initial low-rate charge of ~10 Amps. This continued until they reached a voltage of 76 Volts (1 Volt per cell average), and was followed by three consecutive insolation periods of charging at 30 Amps. Charging was completed during the 4th insolation period using a programmed taper charge. This start-up regime charged the batteries to 100% SOC with a total input of 103 Amp-hrs. Nominal operations were subsequently initiated and battery charge control was provided by the temperature-pressure algorithm.

At beginning of life (BOL), total capacity of the ISS P6 batteries was measured at KSC during IEA final electrical checkout. The battery total capacities during final IEA checkout ranged from 83.0 to 89.9 Amp-hrs when charged using the ISS charge algorithm.

6.0 ISS CHARGE ALGORITHM

The temperature-pressure charge algorithm provides a low-stress charge profile that allows the initial charge current to reach a pre-set maximum and then "tapers" (reduces current) at a rate that is SOC dependent. This profile is designed to maximize the use of available array power, reduce charging stress, and minimize ORU heat generation.

Charge control of this type is necessary in order to ensure orbit-to-orbit energy balance, since power to recharge the batteries varies due to a combination of seasonal orbit conditions:

- User loads
- Extravehicular activity (EVA) operations
- ISS operational scenario (i.e., locked, or non-sun-tracking array mode)

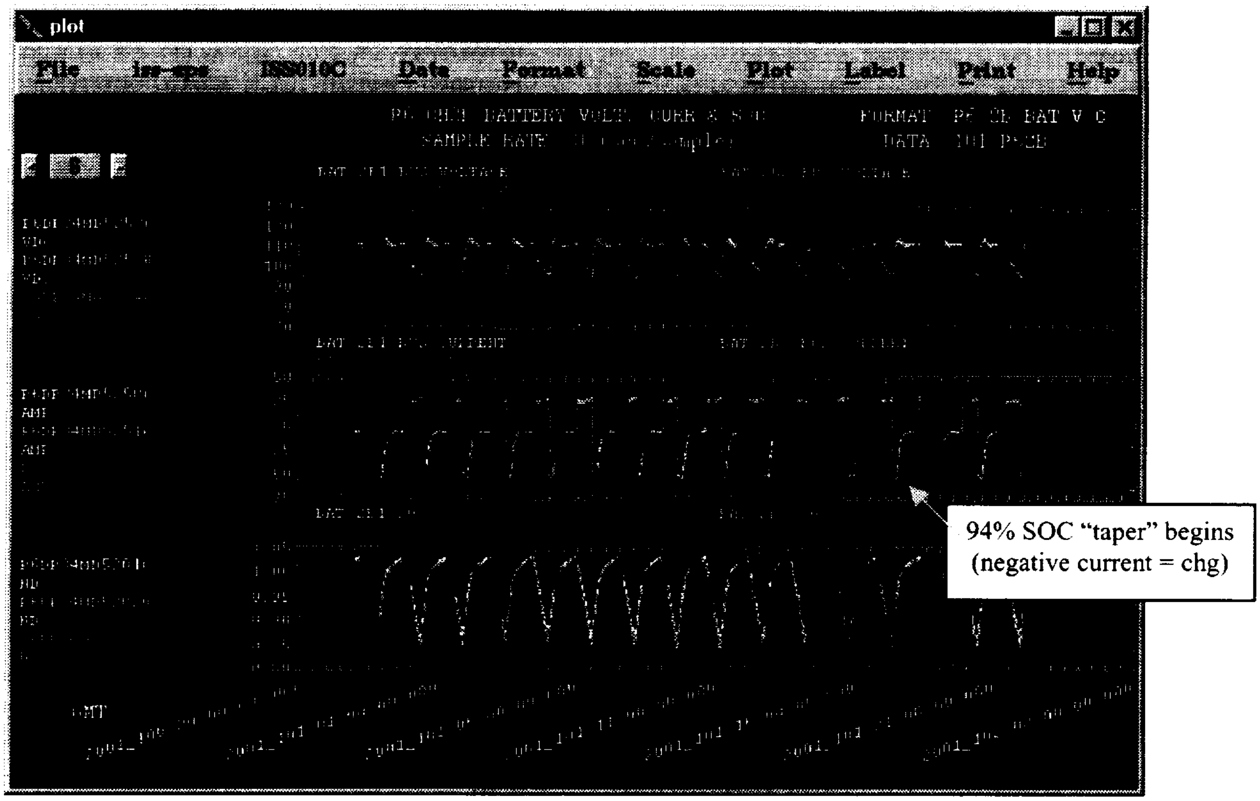

Figure 4: On-Orbit Data Battery Voltage, Current, and SOC

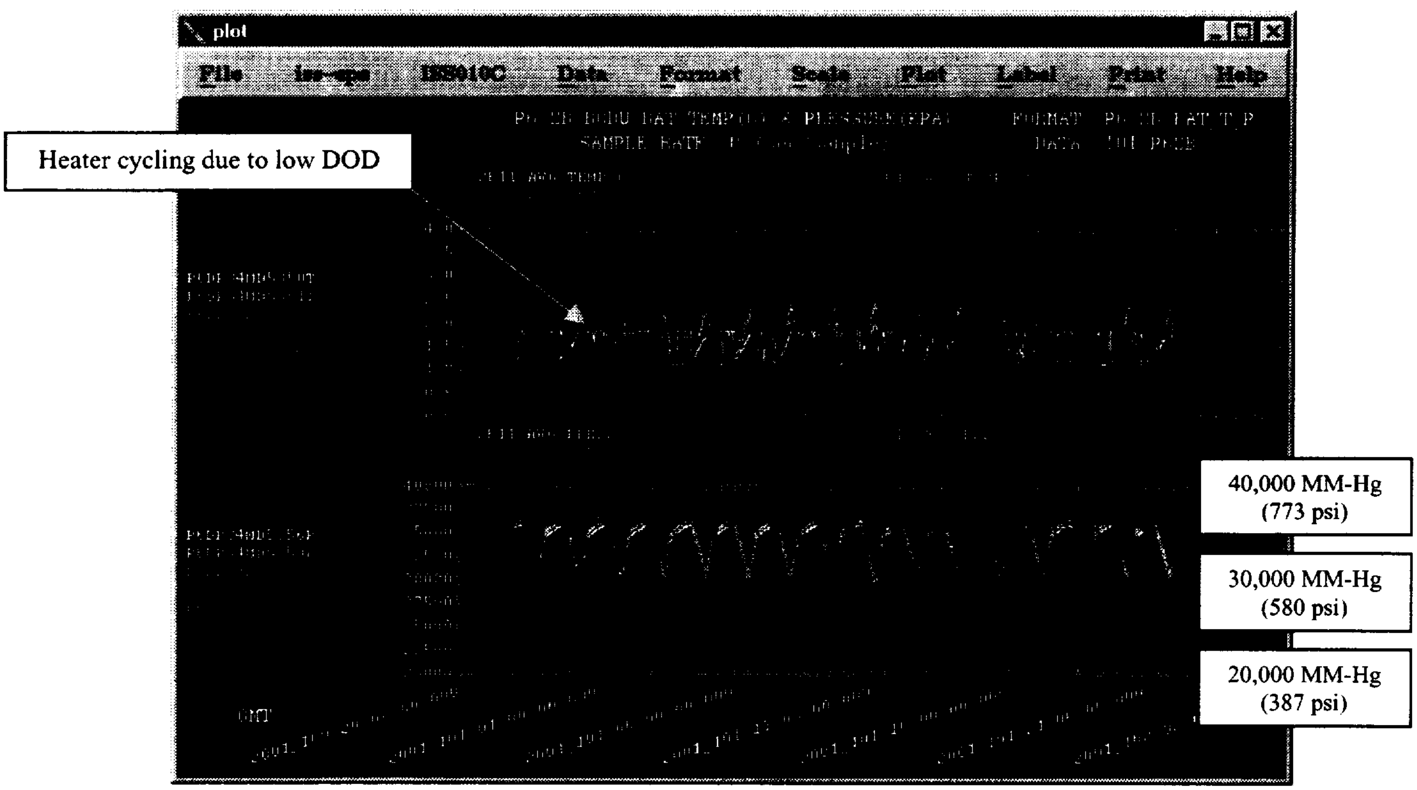

Figure 5. On-Orbit Data Battery Temperature and Pressure

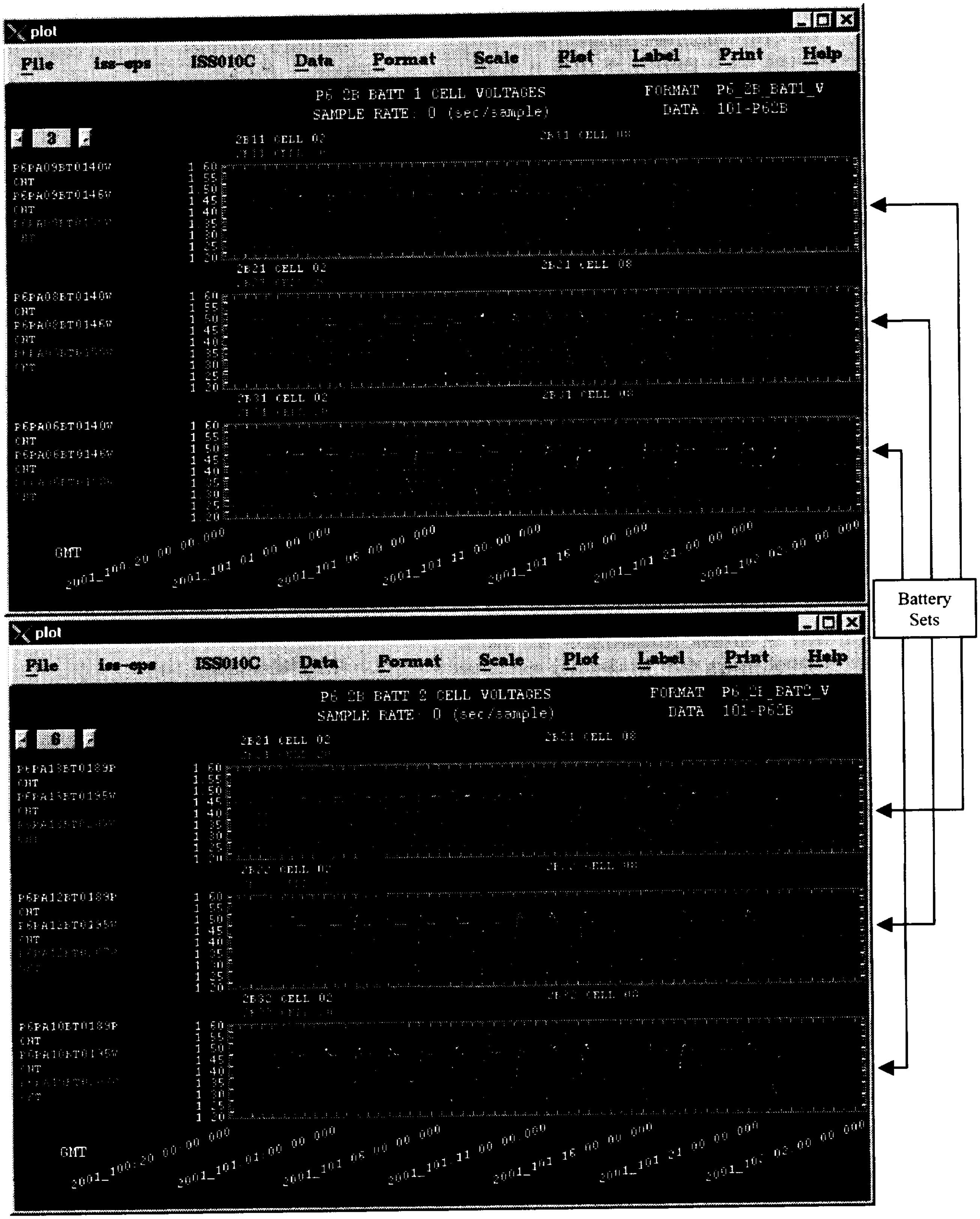

Figure 6: On-Orbit Data, Battery ORU Monitored Cell Voltages (4 cells per ORU 02, 08, 20, and 28)

BOL battery 100% SOC is user set at nameplate capacity (81 Amp-hrs). The charge algorithm calculates SOC using a VanDerWaal's equation and a pressure vs. SOC relationship. Basic or initial parameters taken from battery acceptance data are used to initialize the system before flight. These parameters include strain gauge calibration, initial moles of H2, and pounds per square inch (PSI) per Amp-hr. During LEO operation, the point of recharge where charge efficiency begins to noticeably fall off is 94%. It is at this point where charge current reduction ("taper") begins.

7.0 ISS ON-ORBIT OPERATION

The ISS main power system charge algorithm has pre-set parameters. Maximum charge rate is determined and set based on the on-orbit operation need. Currently, a 50-Amp maximum charge rate setpoint is employed due to operating scenarios that feather arrays to save fuel and/or reduce the possibility of charge build-up on the ISS structure during EVA activity. As such, it is necessary to replenish the battery energy used during eclipse as quickly as possible when it is available from the solar arrays. The taper charge profile is pre-programmed in a look-up table with the following parameters:

SOC%	20	85	90	94	96	98	1.00	1.01	>1.05
Chg Rate (Amps)	50	50	50	50	40	27	10	5	1

The above table is on-orbit programmable and can be revised to allow optimal charge rates for changing operational scenarios, as well as for compensation of changing battery performance characteristics caused by aging.

8.0 ISS ON-ORBIT DATA

The ISS on-orbit data is telemetered to the ground, and is available real time through data screens on console at the Engineering Supports Rooms (ESRs) and the Mission Control Center. Stored, long-term data can be accessed from the Orbiter Data Reduction Complex (ODRC) through the consoles. Representative on-orbit data is shown below in Figs 4, 5, and 6. This data is for Flight Day #101 (April 11 2001). As of this date, the batteries had completed approximately 1,600 LEO cycles. The data depicts the three Channel 2B batteries (6 ORUs). Spaces in the data are caused by data drop-out and are not intentional omissions. The data clearly shows operational ranges of:

- Battery voltage (76 cells) 95 to 115 Vdc
- Maximum charge rate 50 Amps (note that due to ISS EPS conventions, charging current is shown as negative)
- SOC ~85 to ~103% (average DOD 15%)
- ORU temperature range ~1.0 to 2.5°C (Note heater cycling due to ISS operation at less than ORU power design loads)
- Pressure ~580 to ~730 psi
- Cell voltages ~1.26 to ~1.5 Vdc
-

9.0 CONCLUSIONS

The ISS EPS is successfully maintaining power for all on-board loads. This power is currently supplied by six NiH_2 batteries (three per channel) during eclipse. The batteries are designed for a LEO 35% DOD cycle, however, due to the low power demands at this point in the ISS assembly phase, they have been operating at 15% DOD. The batteries are operating nominally and have exceeded all ISS requirements.

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the International Space Station, satellite and aircraft power systems, flywheel technology, spacecraft on-board propulsion, and the 'more electric' technology (MET) insertion in spacecraft, aircraft and launch vehicles.

Power electronic converters are central to the performance of aerospace power systems and spacecraft on-board electric propulsion. Resolution of incompatibility between conventional, 400Hz operating equipment and the variable frequency of MET should promote increased penetration of power electronics into aerospace systems. Future multi-voltage needs and varied load requirements will necessitate the use of multi-voltage level converters. The use of electronic modules with dual-use options and hardware commonality for aircraft and spacecraft should reduce development cost and maximize system re-use, while improving system reliability and performance.

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